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THE EFFECT OF INTERFACE GEOMETRY IN MIXING OF TWO CO-AXIAL SUPERSONIC JETS USING COMPUTATIONAL FLUID DYNAMICS

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ABSTRACT

SCRAMJET is the only air-breathing engine that can achieve hypersonic speeds, accompanied by supersonic combustion. Because of its hypersonic speeds, the residence time of flow in combustor is in the order of milliseconds, which leads to improper mixing and continuous combustion. Thus, combustion has to be associated with holding a flame at high speeds, which makes stable and sustained combustion. To improve the flow mixing and flame holding mechanism in scramjet combustion, the mixing of co-axial jets needs to be analysed and the respective modifications should be made to achieve efficient combustion. The designing and meshing of geometry is carried out using structural meshing software ICEM CFD, before solving the flow in ANSYS Fluent. The analysis is carried for different equivalence ratio of fuel air mixture at various Mach numbers with reacting and non-reacting flows. Mach contours of non-reacting flow predict formation of shock waves and starting characteristics. Mass fraction contours of reacting flow give efficiency of combustion and mixing strength of combustor. CFD model is validated with results of Dual Combustion Ramjet engine.

KEYWORDS: CO-AXIAL, CFD Model & SCRAMJET

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Original Article

1. INTRODUCTION

The recent advances in aerospace vehicle technology, demands the best possible propulsive balance to provide better thrust and minimum drag. Air-breathing engines are the best choice for the propulsion due to its minimum weight. SCRAMJET and RAMJET are the only Air Breathing engines can attain Hypersonic speeds. But these engines cannot start with ground conditions. They require initial supersonic speed to start the engine, which creates shocks to build the pressure required for combustion. As Ramjet can achieve only Mach 6, SCRAMJET is integrated to achieve high hypersonic speeds, which is known as Dual Combustion Ramjet engine (DCR). DCR has two combustors which operate simultaneously; one act as a GG while the other is main thrust producing combustor. For the engine, a portion of the inlet air is diffused to subsonic speed, which is admitted to the Gas Generator (GG) which operates in a fuel rich condition. The hot gases issued from this combustor carry unburnt fuel to supersonic combustor, to which the major portion of the inlet air is admitted through another inlet. Two combustors are coaxial, that is the GG outlet and scram combustor inlets are in the same plane. Hence, the Scram Combustor performance is a function of these jets. The mixing of co-axial jets is governed by the growth of shear layer. Mixing in supersonic shear layer is dependent on the compressibility effect, in addition to the velocity and density across the shear layer.

2. LITERATURE SURVEY

Jong-Ryun Byun, Chul park and Oh Joon Kwon [1] had investigated on Combustor-Isolator in a direct connect DCR experiment. It has an annular isolator, a constant cross-sectional area cylindrical supersonic combustor, and a subsonic-burning gas generator. Mach numbers were varied as 1.78, 1.98, and 2.23 in isolator. The authors are trying to claim wall static pressure distribution measured in the isolator, and the combustor are presented and analyzed to determine the pre-combustion shock train (PCST) length and the wall pressure distributions are derived there from. The experiments were carried out in a direct-connect test facility. The present experimental data have shown that, as the pressure rise increases, the overall length of the PCST remains constant or decreases. This tendency is attributed to the effect that the decrease in the shock-train extension length into the combustor becomes larger than the increase in the length between the beginning of the shock train and the combustor entrance. The extension length has critical value, which is not changed, even when the pressure increases. The test values have shown that the defined shock-train length are less dependent on isolator entrance Mach numbers, at least over the tested Mach numbers. The results also shown that the DCR combustor must be able to operate over a range of equivalence ratios in order to ensure the stability, the equivalence ratio must be sufficiently large to generate secondary combustion securely in the supersonic combustor, at relatively low combustor-inlet Mach numbers a high fuel-equivalence ratio is likely to disrupt the inlet flow.

Tan, J. G., Wu, J. P., and Wang, Z. G [2] had investigated on flow fields and performance of the full-size dual combustor ramjet at Mach4/17km and Mach6/25km flight conditions through direct-connected experiments and numerical simulations. The pressure distributions from simulations are in agreement with that from experiments under both cold flow and hot flow conditions. To investigate reactive flow numerically in DCR combustor, Navier-Stroke (N-S) equations including chemical reaction are solved. Coupled implicit Reynolds Average Navier-Strokes (RANS) equations, re-normalization group (RNG) k- ε turbulence model and finite-rate/eddy dissipation reaction models are adopted, because it has been proved appropriate and valid in various studies; and the study of reactive flow field in laboratory scale ramjet or scramjet combustor Large Eddy simulation (LES) has been used.

Billig, F. S., Waltrup, P. J., and Stockbridge, R. D. [3]a new propulsion concept Integral Rocket Dual Combustion Ramjet (IRDCR) has been described and the volumetric efficiency is improved because of rocket boosters. Combination of subsonic combustion ramjet with scramjet improves engine efficiency and maximum thrust.

Jong ho Choi at al [4]had applieda quasi 1-D model to a supersonic combustor and the variation of temperature and pressure inside combustor were obtained. In addition, the thrust and specific impulse applying fuel regulation by pressure recovery ratio and equivalence ratio were derived.

Tan H et al. [5] has carried out both experimental and computational investigation on the hypersonic inlet for ramjet module of the dual-combustion ramjets (DCR) to obtain the off-design performance and to analyze the internal flow pattern and the restarting characteristics. The results show that the mass flow ratio of the inlet decreases substantially at off-design conditions. Therefore, the paper states to improve the off-design performance of hypersonic inlets andengine have the ability of self-starting at Mach 4.

The literature presents the importance of supersonic combustion and Dual combustion Ramjet in space exploration and its critical areas to be analysed.

3. PROBLEM STATEMENT

Mixing and continuous combustion is the main problem in hypersonic and supersonic speeds. So, the present analysis is carried out for the different flight Mach numbers like 4,5,6 and CFD simulations are done. The corresponding operating Mach numbers will entry with Mach numbers 1.79, 1.98, and 2.23 at isolator. For all the three conditions, both the cold flow analysis (Non-Reacting flow) and hot flow analysis (Reacting flow) had been done, and the CFD simulation results in terms Mach, mass fraction contours are shown in this paper.

4. MODELLING

The primary requirement of CFD simulations is the fluid domain to be analysed. To get the preliminary results and because of computing power limitations, A 2 Dimensional, Dual combustion ramjet is designed in ICEM CFD (a design and meshing module in ANSYS a CAE software) using the geometry shown in figure 1. All the dimensions are in mm. Meshing is major part of CFD procedure. It plays an eminent role in getting accurate results. In spite of ease of mesh generation using unstructured grid, structured grid is generated with blocking to capture shocks and flow pattern in supersonic speeds.

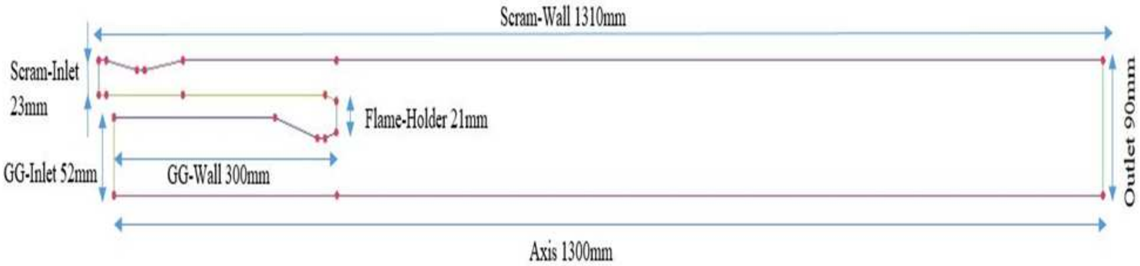

Figure 1: Geometry of DCR Engine.

5. BOUNDARY CONDITIONS

- Kerosene-mixture is used as fuel and the below shown mass fractions are calculated for the equivalence ratio Φ = 1.24.
- K-epsilon (k-∈) turbulence model are derived using renormalization group theory.
- These models are widely used for compressibility and combustion etc.,
- Depending on different equivalent ratios, mass fractions are calculated for all the three Mach numbers 1.79, 1.98, 2.23.

Equivalence ratio$\emptyset = \frac{\left(\frac{\dot{m}_f}{\dot{m}_a}\right)}{\left(\frac{\dot{m}_f}{\dot{m}_a}\right)_s}$

Mass flow rate of fuel $(\dot{m}_f)$=0.162 kg/s

Mass flow rate of oxygen $(\dot{m}_{o_2})$= 1.9275*0.23

$\dot{m}_{o_2} = 0.4433$kg/s

Table 1: Shows Boundary Conditions for Mach 1.79 and Φ = 1.24

Part Names	Mass Flow Rate(kg/s)	Static Pressure (pa)	Total Temperature(k)	Mass Fraction of $C_{12}H_{23}$	Mass Fraction of O_2
GG-Inlet	2.0895	99842.9	925	0.07885	0.2118
Scram-Inlet	5.7825	99482.9	925	---	0.23
Outlet	---	101325	300	---	0.23

The initial conditions for various boundaries of fluid domain are tabulated in table 1. Apart from mentioned conditions no slip conditions are also defined for walls. The initial boundary conditions changes for the Mach numbers 1.98 and 2.23 which has been solved and simulations has been carried out in FLUENT.

6. SIMULATION DONE USING ANSYS FLUENT

Non-Reacting Flow (Cold flow) analysis for different Isolator entry Mach numbers like 1.79, 1.98, 2.23 are done in the Fluent by using above mentioned boundary conditions, by using the k-epsilon turbulence model and kerosene(C12H23), air(O2) mixture is used as species. The following were the results obtained for Cold flow analysis; the range of values in contour was shown in colour map.

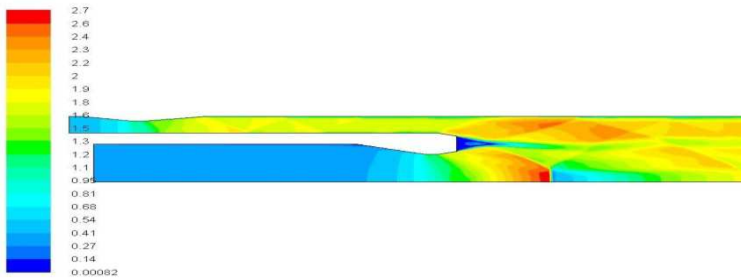

Figure 2: Mach Number Contour of NRF at Mach 1.79 and ER 1.24.

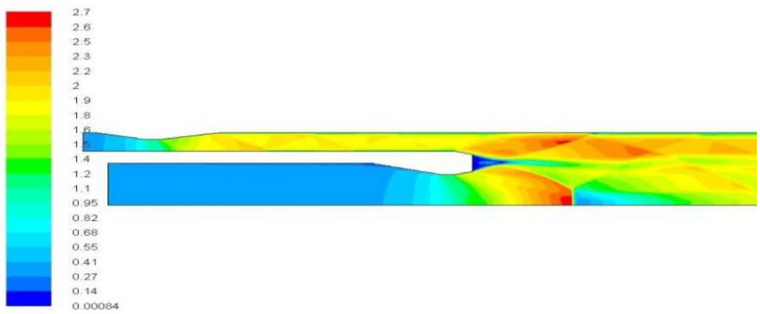

Figure 3: Mach Number Contour of NRF at Mach 1.98 and ER 1.63.

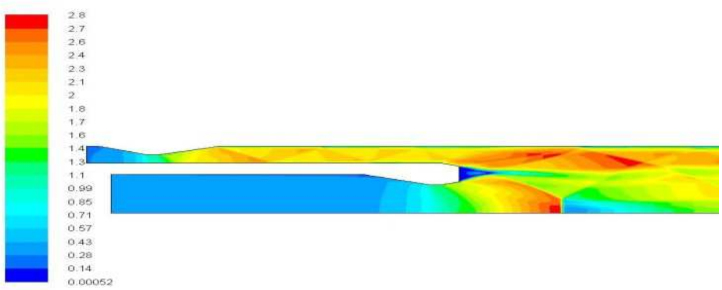

Figure 4: Mach Number Contour for NRF at Mach 2.23 and ER 1.02.

From the above Mach contours, we can observe clearly the formation of pre combustion shock train in the isolator region that is from the distance 110mm to 310mm. Figures 2, 3, 4 shows Mach number contours for the designed isolator entrance Mach numbers of 1.79, 1.98 and 2.23 are matching with the CFD simulations of the validation case, shown in figure 5 below. The strength of shocks increases as Mach number increases. This pre combustion shock increases the pressure and temperature required for combustion.

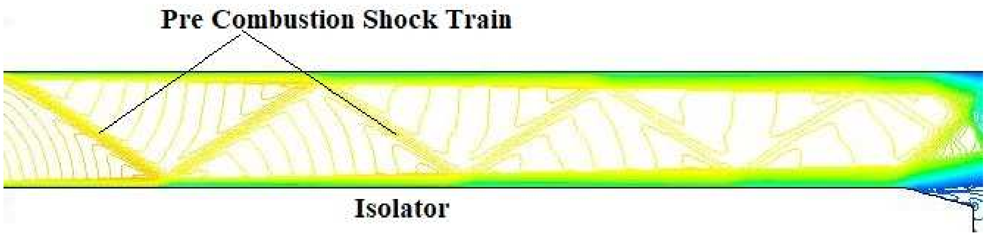

Figure 5*:* PCST in Isolator Duct

Isolator is an essential component of a DCR engine. With combustion, a pressure gradient is imposed on incoming supersonic flow and due to presence of boundary layer, a series of shocks called a ‘shock train’ also referred here as ‘Pre-combustion Shock Train’ (PCST) forms, and required pressure rise occurs gradually over a length. This shock structure can move forward in the inlet, disrupting the inlet function. This can cause failure of the engine. The isolator is designed to contain this shock train, preventing it from unstarting the inlet. A PCST has several advantages for flame stabilization:

It increases the static temperature and pressure of the incoming air flow, thus reducing the Ignition delay.

It decelerates the flow, and thus increases the residence time of the air/fuel mixture inside the combustor.

Reacting Flow (Hot flow) analysis is done with the continuation of cold flow after the flow has stabilized in the engine; volumetric has been started in the species transportation four step chemical reactions have been used foe combustion.

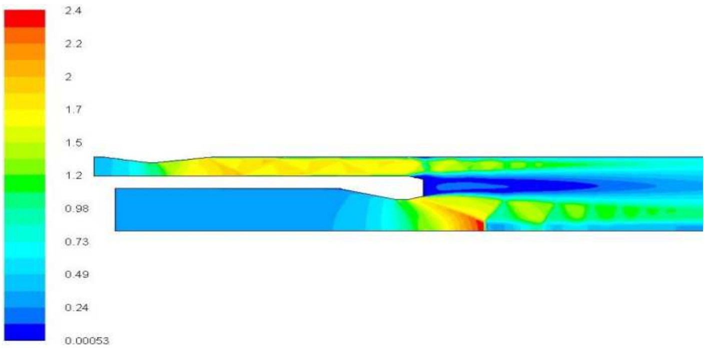

Figure 6: Mach Number Contour for RF at Mach 1.79 at ER 3.05.

The above Mach contour predicts the formation of shock waves decreasind Maximum Mach 2.4 to 1.7, followed by normal shock with downstream Mach of 0.7.

From mass fraction contour, we can see clearly the mixing in combustion chamber has improved when compared to ER 1.24 and 1.63. As ER is increased, the mixing of gases is been improved and velocity has been gradually reduced. As Er increases, this causes reduction in thrust force for the engine.

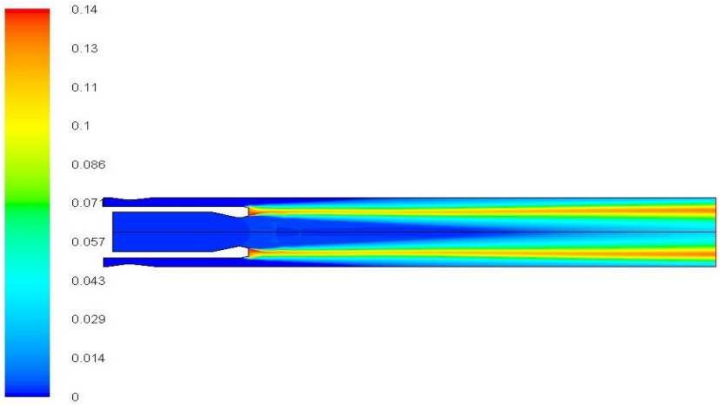

Figure 7: Shows Mass Fraction of co2 for RF at 1.79 at ER 3.05.

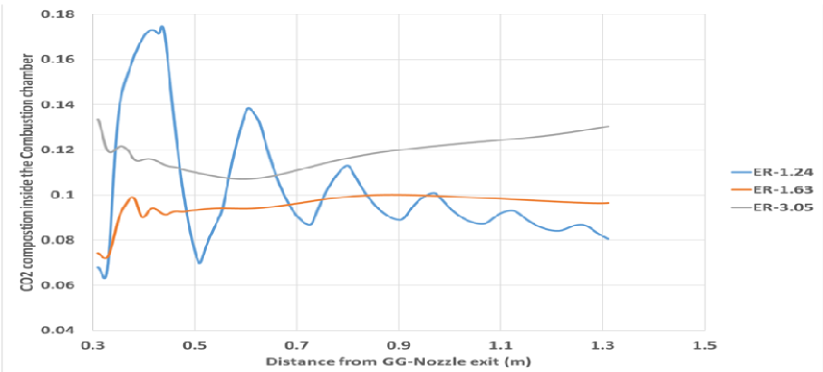

Figure 8: Mass Fracion of CO_2 RF at different ER 1.24,1.63 and 3.05.

CFD simulation results of reacting flows are analyzed by a graph about distribution of mass fraction of combustion products CO_2 along DCR. The following points are accumulated from the above graph.

- The formation of CO_2 is high at the exit of GG for the ER 1.24, and it has been gradually reduced at the end of the combustor chamber.
- The formation of CO_2 is slightly increased at the exit of GG for the ER 1.63, and it has maintained almost equal mass fraction till the exit of Combustion chamber
- It is reverse for the case of ER 3.05. initially it is reduced, there after the mass fraction has increased and had formed the highest product of CO_2.
- Depending on the highest mass fraction, we can give a statement that combustion percentage is high.
- In all the three ER from the mass fractions of $CO_{2,}$ we can say that 3.05 has the amount of CO_2 product formed, when compared to other two ER.
- Mass flow rate increases with the increase in the ER.
- Formation of apex and mixing mechanism can be clearly observed as the ER increases, the other way to express this is rich fuel air mixture has better mixing when compared to lean air mixture.

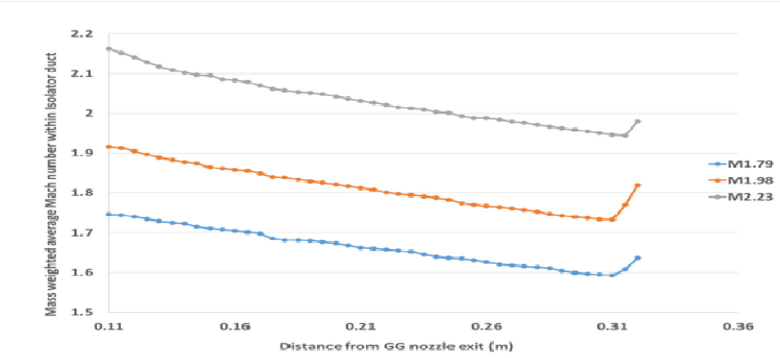

Figure 9: Mass-weighted Average Mach Numbers with in the Annular duct for the Designed Isolator Entrance Mach Numbers of 1.79, 1.98, 2.23.

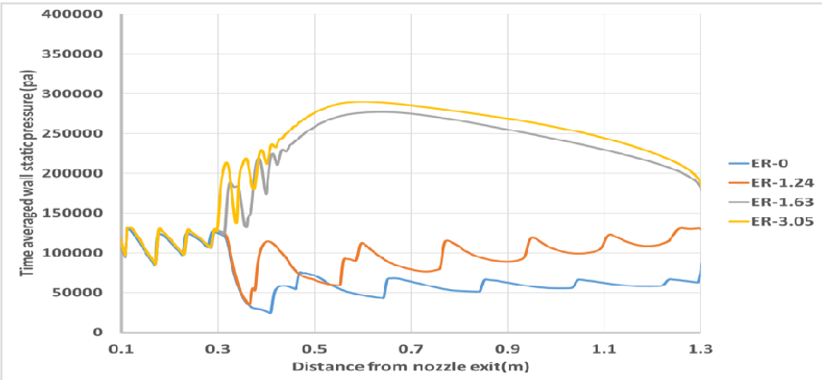

Figure 10: Time-averaged wall pressure Distribution for different Equivalence Ratios at Mach = 1.79 at 0° inlet Angle.

- Pressure rise in the isolator region is almost same for different equivalence ratio.
- For Φ=0 i.e., no fuel condition there is pressure drop in the supersonic combustor.
- For Φ=1.24, there is increase in static pressure along the length of the combustor.
- For Φ=1.63 and Φ=3.06, that is rich fuel mixture, in this condition is there is sudden rise of static pressure in supersonic combustor.
- As increase in the equivalence ratio, there is increase in the static pressure and in the higher Φ values, the combustion efficiency is high.

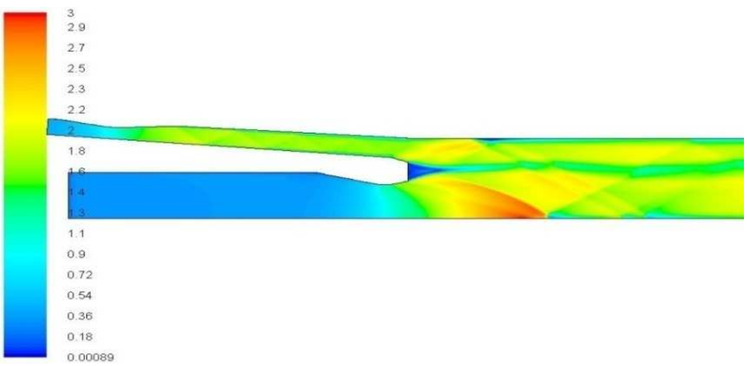

Figure11: Mach Number Contour for Mach 1.79 at 5° Isolator inlet Angle.

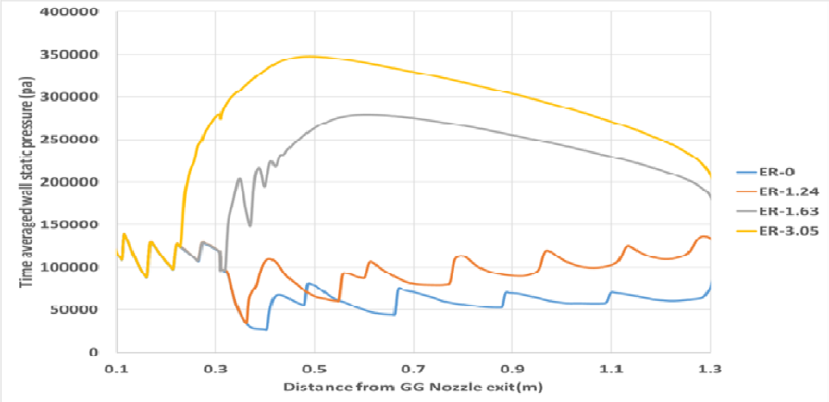

Figure 12: Time-averaged Wall Pressure distribution for different Equivalence Ratios at Mach1.79 at 5˚ Inlet Angle.

- Pressure drop in the isolator is high at 5˚ when compared to 0˚.
- As incidence increases, pressure rise in the combustion chamber increases.
- Pressure in the combustor has been increased and mixing has been improved for 5˚

7. CONCLUSIONS

For this thesis, many simulations have been carried out for the 2D Axi-symmetry geometry, and we have concluded that the mixing mechanism is better for ɸ 1.63 is among the different equivalence ratios, 5˚ is better when compared to other angles. From the apex formation, mixing mechanism has been observed, we can notice apex for 0˚ isolator change at the end of the supersonic combustion chamber, whereas, we can observe the apex formation has been moved to the center of the supersonic combustion chamber for the change of 5˚ isolator angle. Furthur, there was no improvement for the 10˚ isolator angle, and we can conclude that at 5˚ isolator angle, we had better mixing mechanism. For a better combustion and mixing of flows, the equivalence ratio should be increased.

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